

TRANSLUNAR ABORT TECHNIQUES
FOR NONFREE-RETURN MISSIONS

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ABSTRACT

A method of translunar abort for nonfree-return missions is considered in which the service module is jettisoned (more than doubling the abort propulsion capability of the lunar-module descent propulsion system) before the transearth maneuver is performed. During the transearth flight, the lunar module provides the life-support and power requirements for the crew.

CONTENTS

Section	Page
SUMMARY	1
INTRODUCTION	1
ABORT VELOCITY REQUIREMENTS	2
SINGLE-IMPULSE ABORT	3
MULTI-IMPULSE ABORT	3
ABORT TECHNIQUES	4
RESULTS	6
CHANGES TO THE LUNAR MODULE	7
CONCLUSIONS	7
REFERENCE	8

TABLE

Table		Page
I	WEIGHT SUMMARY	9

FIGURES

Figure		Page
1	Profile of a single-impulse abort trajectory after an SPS engine failure at LOI	10
2	Single-impulse abort ΔV requirements for a maximum-payload trajectory to Copernicus	11
3	Single-impulse abort ΔV requirements of nonfree-return TL trajectories to Aristarchus and Copernicus	12
4	Multi-impulse abort ΔV requirements of nonfree-return TL trajectories to Copernicus	13
5	Profile of multi-impulse abort after engine failure at LOI	14
6	Abort ΔV available from LM DPS	15
7	Service-module separation prior to LM DPS abort propulsion maneuver	16
8	Command-module DPS abort maneuver onto a high-speed TE trajectory following SM separation	17
9	Flight profile for abort with SM jettisoned to increase ΔV capabilities	18
10	Abort ΔV requirements for high-speed TE trajectories, Copernicus mission	19
11	Abort ΔV requirements for high-speed TE trajectories, Aristarchus mission	20
12	Lunar-module electrical energy consumed to provide crew emergency life support after SM separation	21
13	Lunar-module O_2 and H_2O consumed to provide crew emergency life support after SM separation	22

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SUMMARY

Nonfree-return trajectories are being considered for advanced lunar missions. In the event of a command-and-service-module engine failure, a method of providing abort capabilities from low-energy, nonfree-return translunar trajectories by using the lunar-module descent propulsion system is needed. Abort velocity requirements are given which indicate that for low-energy translunar trajectories, the descent propulsion system does not have sufficient propulsion capability to abort the entire spacecraft. An abort technique is examined in which the service module is jettisoned, and the crew is transferred to the lunar module, with emergency life support and power provided by the lunar-module systems. The command module is retained for reentry. The resultant decrease in spacecraft weight enables the descent propulsion system to abort the spacecraft from the low-energy translunar trajectories and to return the spacecraft to earth before the life-support capabilities of the lunar module are exceeded. The abort maneuver is delayed as long as possible (8 to 20 hours) to reduce the time required for lunar-module life support and to obtain earth-landing longitude control. The proposed technique requires little or no systems modifications.

INTRODUCTION

The enlarged payload and performance requirements of longer and more complicated lunar-landing missions will require the use of nonfree-return translunar (TL) trajectories. On a mission to the crater Copernicus, for instance, a 110-hour nonfree-return TL trajectory allows the spacecraft to deliver 5000 to 6000 pounds more payload into lunar orbit than is possible with a free-return TL trajectory. The 5000 pounds could be used for any one, or combinations, of the following: (1) a larger lunar module (LM), (2) an orbit-science package, and (3) increased command-and-service-module (CSM) performance such as CSM assistance of the LM for lunar orbit maneuvers. In free-return TL trajectories, the CSM barely has the capability of delivering an Apollo LM (which has only a 22-hour surface lifetime) to a 60-nautical-mile orbit, and this capability is for only a limited region of the moon. With nonfree-return TL trajectories, a large variety of mission possibilities becomes available.

The advantage of the free-return TL trajectory is the free-abort capability in case of service-propulsion-system (SPS) failure at lunar orbit insertion (LOI). However, this advantage is lost as soon as LOI begins, and an SPS failure after a partial LOI burn is almost as difficult an abort problem with a free-return TL trajectory as with a nonfree-return TL trajectory.

If nonfree-return TL trajectories are to be used, a technique must be found to abort the mission in case of a complete SPS failure at LOI. The only other propulsion source available is the LM descent propulsion system (DPS) which is limited in propulsion capabilities when maneuvering the fully fueled CSM. The LM ascent propulsion system (APS) cannot be used because it does not have sufficient control capabilities when attached to the CSM.

Studies of single- and multi-impulse abort velocity ΔV requirements show that for the 110-hour TL trajectory to Copernicus, the minimum ΔV required for abort is approximately 2700 fps. At the time of the abort, the SM contains almost 40 000 pounds of unusable SPS fuel. With so much extra inert mass, the DPS can produce only 1950 fps of abort ΔV . This is far short of the required 2700 fps. There is no provision for emptying the SPS fuel tanks, and the tanks cannot be dropped off, since the SM fuel tanks and the CM life-support system are part of the same structure in the SM. One solution is to jettison the entire SM, including its fuel and life-support system; to abort onto a high-speed, single-impulse transearth (TE) trajectory with the DPS (which can generate 4600 fps with the reduced weight); to power down the CM; and to live in the LM during the TE phase. The CM can be powered up and the LM jettisoned immediately prior to reentry.

In the following sections of this report, abort ΔV requirements are presented, and the feasibility of abort from nonfree-return TL trajectories, using the LM for life support over a portion of the return trajectory, is examined.

ABORT VELOCITY REQUIREMENTS

At the beginning of the LOI burn, the spacecraft is composed of the CSM (with nearly full fuel tanks), which is docked to the LM (with full fuel tanks). The weights of these vehicles are given in table I. It is assumed that a total SPS failure occurs at initiation of the LOI burn; that is, the TL trajectory is of the low-speed, nonfree-return type in which, with no previous warning, the SPS engine fails to start. The problem is to find a way to return the crew to earth from this point.

Translunar trajectories for lunar-landing missions to the craters Copernicus and Aristarchus are used as reference trajectories for abort ΔV data. The launch dates are May 28, 1971, for the Copernicus trajectories and July 28, 1971, for the Aristarchus trajectories. These are the launch dates giving optimum-performance to the respective landing sites.

SINGLE-IMPULSE ABORT

In a single-impulse abort, the spacecraft makes a single burn directly from the TL coasting trajectory onto a TE trajectory (fig. 1). With a single-impulse abort trajectory, the required ΔV rises sharply with an increase in the time between pericynthion passage (at which the SPS failure occurred) and the application of the abort impulse. This rise is shown in figure 2 for the Copernicus lunar-landing mission. The required ΔV reaches a peak and levels off to a nearly constant value at a time past pericynthion of slightly less than 2 hours. Because 2 to 4 hours are required to ready the LM descent stage for abort, the lower ΔV aborts immediately after pericynthion are not, in practice, available. The ΔV requirements also increase with an increase in TL flight time (fig. 3). The lower speed, nonfree-return TL trajectories have the greatest payload capabilities, but present the most difficult abort problem. In few cases of interest is the required ΔV within the 1950-fps capability of the DPS when it is moving the fully loaded spacecraft. The ΔV requirements are usually between 2500 and 3500 fps.

MULTI-IMPULSE ABORT

Because of the high ΔV required for single-impulse aborts, multi-impulse aborts were studied by using the n-impulse program developed in reference 1. The program, as used in this study, generated the optimum n-impulse transfer between two given state vectors. The program also generated the optimum number of impulses n. Figure 4 is a summary of those results of the study which are of interest in this report. The minimum multi-impulse ΔV found for an abort from the 110-hour TL trajectory is greater than 3000 fps if the 2-hour delay required after pericynthion to ready the LM is considered.

All of the most desirable multi-impulse abort solutions had similar profiles (fig. 5). The first DPS burn occurs as soon after LOI failure as possible and places the spacecraft into a high ellipse around the moon. A ΔV of approximately 600 fps is required for an ellipse with a 10 000-nautical-mile apocynthion. Thus, the spacecraft is inserted into a capture orbit so that lunar gravity can swing the velocity vector around, eliminating the need for the engines in making large flight-path angle changes. A second impulse is generated at apocynthion to provide for any plane change that may be required. The third impulse, which is the second major burn, is generated slightly before pericynthion and injects the spacecraft onto the TE trajectory defined by the final state vector. A fourth impulse (usually small) is normally generated at the position of the final state vector to give the exact trajectory to return to earth.

The intermediate orbits with greater apocynthions require less propulsion capability for maneuvering in and out of orbit. However, the higher orbits also have a longer period, and the time between LOI failure and spacecraft exit of the lunar sphere of influence (SOI) increases. The position of the final state vector must be rotated in a posigrade direction corresponding to the 13-deg/day apparent revolution of the earth around the moon. The change in direction shifts the best firing position for the TE injection impulse farther and farther from pericynthion, making the injection increasingly

costly in propellant. These two trade-offs result in optimum ellipses with apocynthions of approximately 13 000 nautical miles and periods of approximately 3 days.

The n-impulse program used in the previously discussed study gives the best transfer between two specified state vectors for a given time interval Δt . The program was capable only of generating impulses for operations around a single body; therefore, it was necessary to obtain the initial and final state vectors from other sources. Initial state vectors (at the time of abort) were obtained for TL trajectories to Copernicus with different flight times. The final state vector was the state vector at the lunar SOI, obtained from the best single-impulse TE trajectory starting from the lunar parking orbit for a Copernicus landing and rotated to account for the apparent revolution of the earth during the abort interval. This TE trajectory required a flight time of approximately 100 hours. Other TE initial state vectors were examined, and the differences in ΔV were not found to be significant. However, all final state vectors which were examined were from the same type of low-energy, 100-hour TE trajectories; and subsequent single-impulse studies have indicated that higher speed trajectories may give better results. The use of higher speed TE trajectories increases the change in absolute velocity (energy state), but also decreases the required change in the flight-path angle. However, it has become apparent that while better trajectory solutions may be found, the multi-impulse technique is not a panacea that will solve the abort problem. In addition, a different approach has been found (which is discussed in the next section of this report); therefore, further n-impulse studies have not been undertaken.

ABORT TECHNIQUES

Each of the two abort modes which were examined requires an abort ΔV on the order of 3000 fps for the low-speed, low-energy TL trajectories which result in greater payload delivery capability. The ΔV available for the abort using the LM DPS is given in figure 6. The ΔV is shown as a function of the CSM main-engine (SPS) fuel remaining on board. None of the SPS fuel is usable for the abort; it is deadweight. For a fully loaded spacecraft, the DPS can generate only a 1950-fps ΔV . Thus, with the type of low-speed, nonfree-return trajectories necessary for the payloads and orbit orientations of advanced lunar missions, the DPS is normally unable to perform the direct abort with the fully loaded spacecraft that had been previously envisioned.

The unusable propellant in the SPS composes approximately one-half of the mass remaining after the abort burn and reduces the available ΔV so much that an abort cannot be made. The SPS fuel is completely useless, since the SPS engine does not work. If the propellant could be jettisoned, the increase in available ΔV would enable the spacecraft to abort from almost any TL trajectory.

No current provision for jettisoning the SPS fuel exists, and it would apparently be difficult to make the required modifications. The events that would probably cause SPS failure with hypergolic fuels would also prohibit emptying the fuel overboard, since either pumping the fuel into the firing chamber or pumping the fuel outside is essentially the same problem.

If the fuel could be dumped overboard, the toxicity and combustibility of the hypergolic propellant might create additional problems. In any case, hardware modifications of an extensive nature would be needed to jettison the fuel.

If the entire SM is jettisoned, the ΔV made available by using the LM DPS increases to 4600 fps. This technique has not been considered before because the main mission life-support and power supplies are in the SM, with the exception of those placed in the CM for the reentry phase. However, the crew could transfer to the LM and use LM life support and power. The CM would then be powered down until it is needed for reentry. The spacecraft, during and after impulse, would be the LM, with the CM being carried along as a reentry vehicle (figs. 7 and 8).

Once the SM is jettisoned, time becomes a critical factor because of the limited life-support capabilities of the LM. Because of this limitation, the spacecraft is allowed to coast to the vicinity of the lunar SOI (i. e., 30 000 n. mi. from the moon) before the abort impulse is applied, thus minimizing the distance remaining to be traveled after the burn (fig. 9). By the time the lunar SOI is reached, the lunar gravity has slowed the spacecraft to the point that the relative positions of the earth, moon, and spacecraft change very slowly; and an additional delay of 10 to 20 hours at this point has little effect upon either the ΔV or flight time after the impulse. The spacecraft can then obtain control of earth-landing longitude by varying the delay before the impulse rather than by varying the flight time after the impulse.

In figure 10, the TE flight time after the abort impulse is shown as a function of the ΔV required at the impulse, for an abort from a 110-hour trajectory to Copernicus. This TE flight time is the total time that the LM systems must maintain a life-support capability. Two curves are shown: one curve is for the impulse occurring 8 hours past pericyynthion ($\Delta t = 8$), and the other curve is for an impulse occurring 14 hours past pericynthion ($\Delta t = 14$). The minimum time of flight is almost the same in both cases, approximately 42 hours. The result is that, as expected, the time of the SM jettison and abort impulse can be varied so that a suitable landing site will be in the correct position for the reentry without an increase in the total time from impulse to reentry.

Figure 11 is similar to figure 10, but is for an abort from a trajectory to Aris-tarchus. The minimum time between impulse and reentry, the time the LM must maintain a life-support capability, is approximately 42 hours for both missions. The ΔV curves are very steep in this region, which indicates that 42 hours is a solid limit and that a large increase in available ΔV is necessary to significantly reduce the minimum time required for LM life support.

The usage of critical consumables was investigated for the LM in deep space with three men on board. This investigation revealed that there is apparently no major problem involved with three men living in the standard LM on an emergency basis for the 45 hours (including warmup) required of the high-speed abort. However, there is an apparent problem with the CM electrical power.

At CM/SM separation, 3480 W-hr are available from the CM batteries. During the CM powered-down phase, all consumers of power in the CM can be turned off except for the computer complex. The resultant power level, 53 watts, is low, but in the 42 hours of the powered-down phase, the total drain is 2226 W-hr, or considerably

more than one-half of the available CM energy. If this energy were to come from the CM power sources, they would be completely depleted about the time the reentry parachute opens. No power would be left to upright the CM if it landed apex down, nor would any power be left for beacon operations. The beacon operations would be necessary if the landing should occur out of sight of the recovery forces.

A CSM/LM umbilical connection is provided between the power sources on the CSM and LM, and the CM power level of 53 watts is low enough to be easily supplied through the connection by the LM. The LM batteries have a total energy greater than 60 kW-hr; therefore, supplying the 2 kW-hr needed by the CM would not significantly shorten the LM power lifetime. However, a relay system currently disconnects the power lead of the umbilical when the descent stage is attached. A method of circumventing this relay should be incorporated so that the descent batteries can supply the CM power requirements during the TE coast. With this change, the length of the TE flight time is limited only by the LM power sources, which can power the LM and the CM for 57 hours (fig. 12).

The depletion rates of the other two major consumables, oxygen and water, are shown in figure 13. The oxygen could last as long as 62 hours, while the water would be depleted in 61 hours. In neither of these cases does the addition of another crew-member significantly alter the results. Most of the oxygen is lost in spacecraft leakage, while the water is consumed in vehicle cooling.

The availability of sufficient lithium hydroxide (LiOH) in the LM was also briefly examined and is not a major problem for the high-speed abort.

There are two primary canisters for the LiOH, which is used to remove exhaled carbon dioxide (CO_2) from the air supply. One canister is stored in the LM cabin; the other is stored in the descent stage and, in the nominal landing mission, is retrieved during the first lunar-surface extravehicular-activity period.

Each primary canister can remove the CO_2 generated during approximately 40 active man-hours. However, during the abort, the crew would be inactive; therefore, the CO_2 generated by the three men should not be significantly greater than the CO_2 produced by one active man. The one onboard primary canister might thus be sufficient for the abort. In addition, however, there are five usable secondary LiOH cartridges in the cabin with a combined capacity of from two to three times that of one primary canister. The availability of this secondary LiOH supply gives a total onboard supply of from two to four times what should be required during this type of abort.

RESULTS

Single-impulse and multi-impulse aborts from low-speed, nonfree-return TL trajectories were examined. Abort ΔV requirements in both abort modes proved to be approximately 3000 fps, which is much more than the 1950-fps capability of the LM DPS when moving the massive CSM/LM spacecraft configuration remaining after an LOI burn failure. The abort technique developed in this study was to jettison the SM

before the abort burn. This maneuver reduces the total mass of the spacecraft, and the DPS can then provide a 4600-fps ΔV .

The CSM consumables are in the SM, which is left behind; therefore, the crew must transfer to the LM and depend on the LM life support. The LM life-support capabilities are sufficient if the time spent between jettisoning of the SM and reentry at earth is less than 57 hours. The spacecraft, with the SM still attached, is allowed to coast out to the region of the lunar SOI (approximately 30 000 n. mi.). The SM is then jettisoned, and the spacecraft aborts onto a high-speed TE trajectory. With the 4600-fps propulsion capability that is now available, the minimum TE transfer time after the abort burn will be 40 to 45 hours. Emergency life support can easily be maintained on board the LM for this period of time and, in fact, could be provided for as much as 57 hours, which gives a performance pad and contingency capability. The maximum reentry speeds for the TE trajectories were less than 36 400 fps. The current CM heat shield is capable of reentry speeds as great as 36 600 fps.

The minimum TE transfer time of 42 hours appears to be independent of the original TL trajectory. Consequently, this abort technique should be suitable for all the landing sites now being considered.

CHANGES TO THE LUNAR MODULE

Apparently, only two modifications are necessary to the LM for this abort technique. First, the relay system involved with the umbilical cord between the CM and LM must be altered so that power can be sent back to the CM from the LM while the descent stage is attached. This requirement was discussed in detail in an earlier section of this report. Second, a third suit umbilical outlet should be installed in the LM for the third crewmember in the event of LM pressure loss.

CONCLUSIONS

An abort technique has been described in which the abort propulsion capability of the lunar-module descent propulsion system is more than doubled by jettisoning the service module and shifting life-support requirements to the lunar module. The 4600-fps ΔV available with this technique is enough to provide abort capability in the event of spacecraft-propulsion-system failure at lunar orbit insertion for all normal translunar trajectories, including the low-speed, nonfree-return trajectories which result in maximum payload capabilities.

The length of time that life support must be provided by the lunar module is determined by the minimum time interval between the abort ΔV and earth reentry. The minimum time proved to be similar (approximately 42 hours) for both sites examined (Copernicus and Aristarchus), even though the sites represent a wide spread in orientation of the original translunar trajectories. The time is within the 57-hour life-support capability that the current lunar-module systems can provide. There are indications that this relative invariance in life-support requirements will be true for trajectories to a wide region of the lunar surface. Furthermore, the post-Apollo lunar

modules will probably have increased life-support capabilities; consequently, the combination of high- ΔV capability with long transearth flight times should result in the capability to abort from any translunar trajectories that are likely to be of interest for some time. A safe return to earth in the event of a service-propulsion-system engine failure at lunar orbit insertion can thus be effected by the lunar-module descent propulsion system, even for low-energy translunar trajectories. This capability permits the maximum-payload trajectories to be used in the mission-planning process without relinquishing the abort constraints.

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Houston, Texas, February 17, 1969
981-20-00-00-72

REFERENCE

1. Jezewski, Donald J.; and Rozendaal, Harvey L.: An Efficient Method for Calculating Optimal Free-Space N-Impulse Trajectories. AIAA Journal, vol. 6, no. 11, Nov. 1968.

TABLE I. - WEIGHT SUMMARY

Block II CM (manned), lb	13 000
Block II SM (including reaction control system and unusable SPS fuel), lb	11 560
Usable SM propellant, lb	39 740
Total LM weight (unmanned but including full fuel), lb	33 500
Total spacecraft weight, lb	97 800
Usable LM DPS propellant, lb	17 510
DPS engine specific impulse I_{sp} , sec	299.4

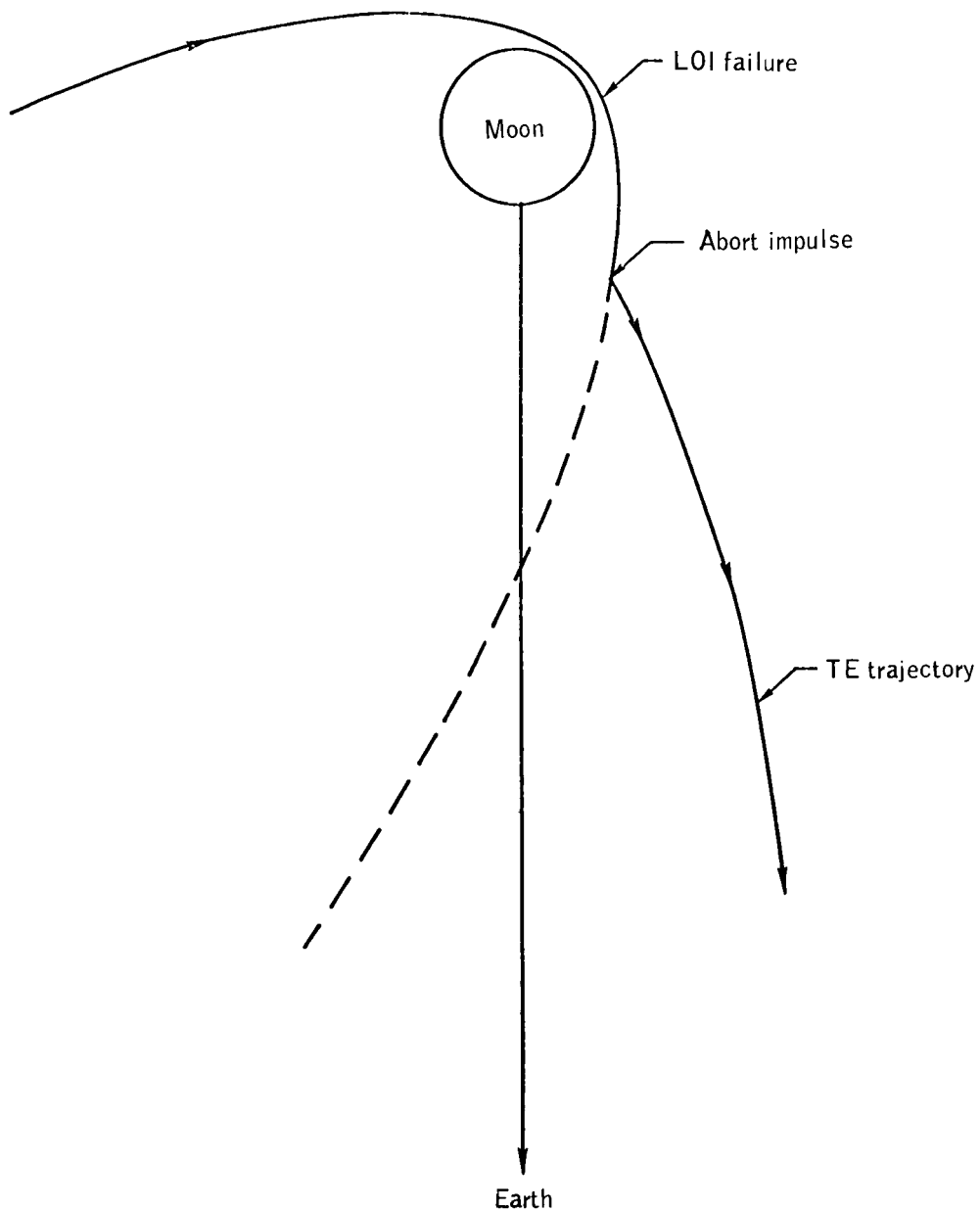


Figure 1. - Profile of a single-impulse abort trajectory after an SPS engine failure at LOI.

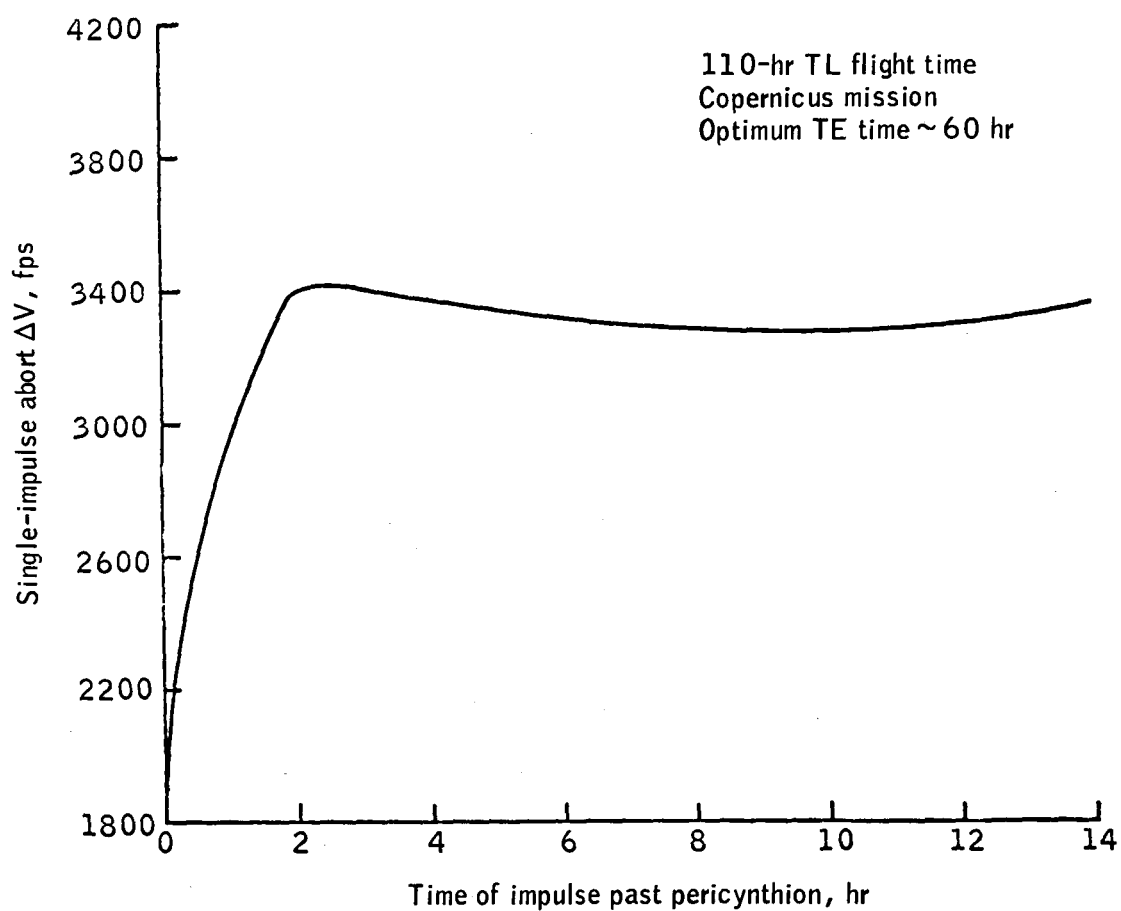


Figure 2. - Single-impulse abort ΔV requirements for a maximum-payload trajectory to Copernicus.

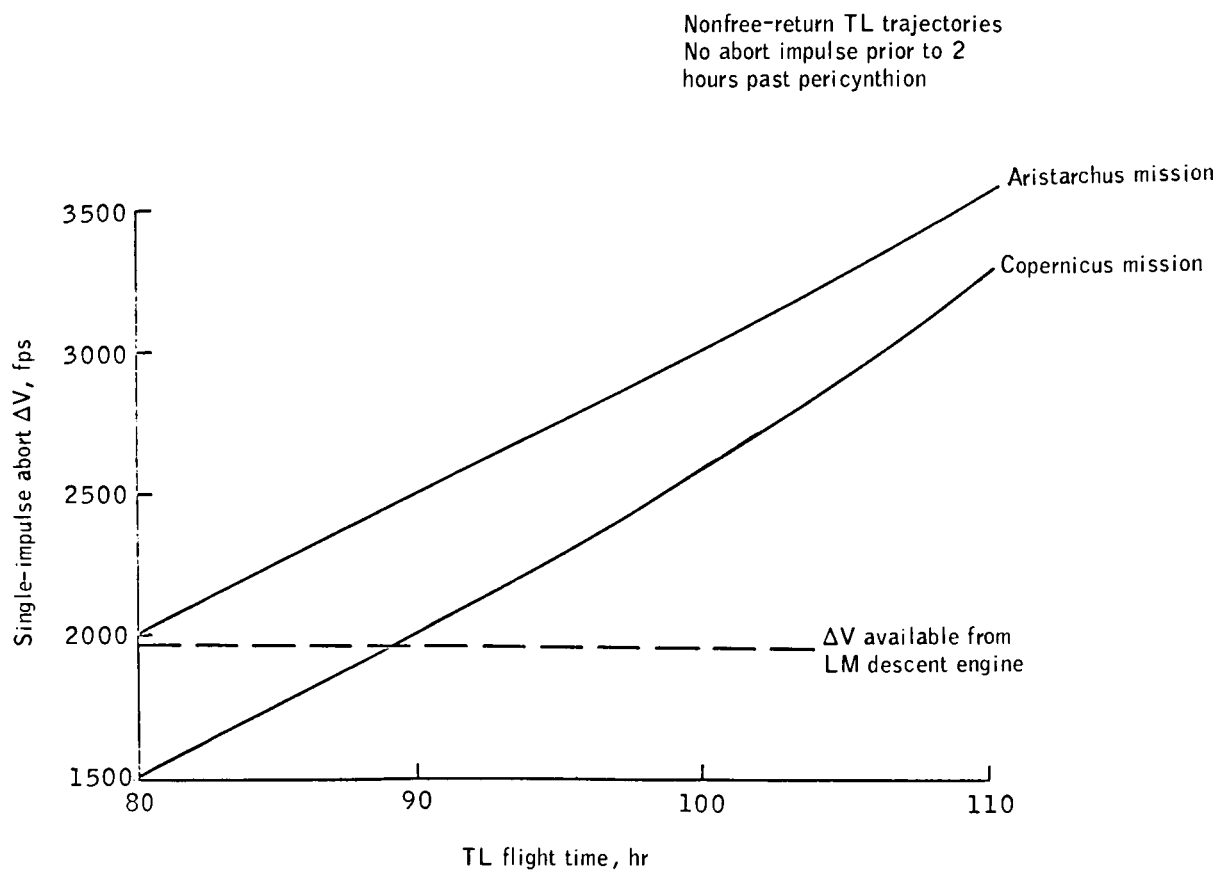


Figure 3. - Single-impulse abort ΔV requirements of nonfree-return TL trajectories to Aristarchus and Copernicus.

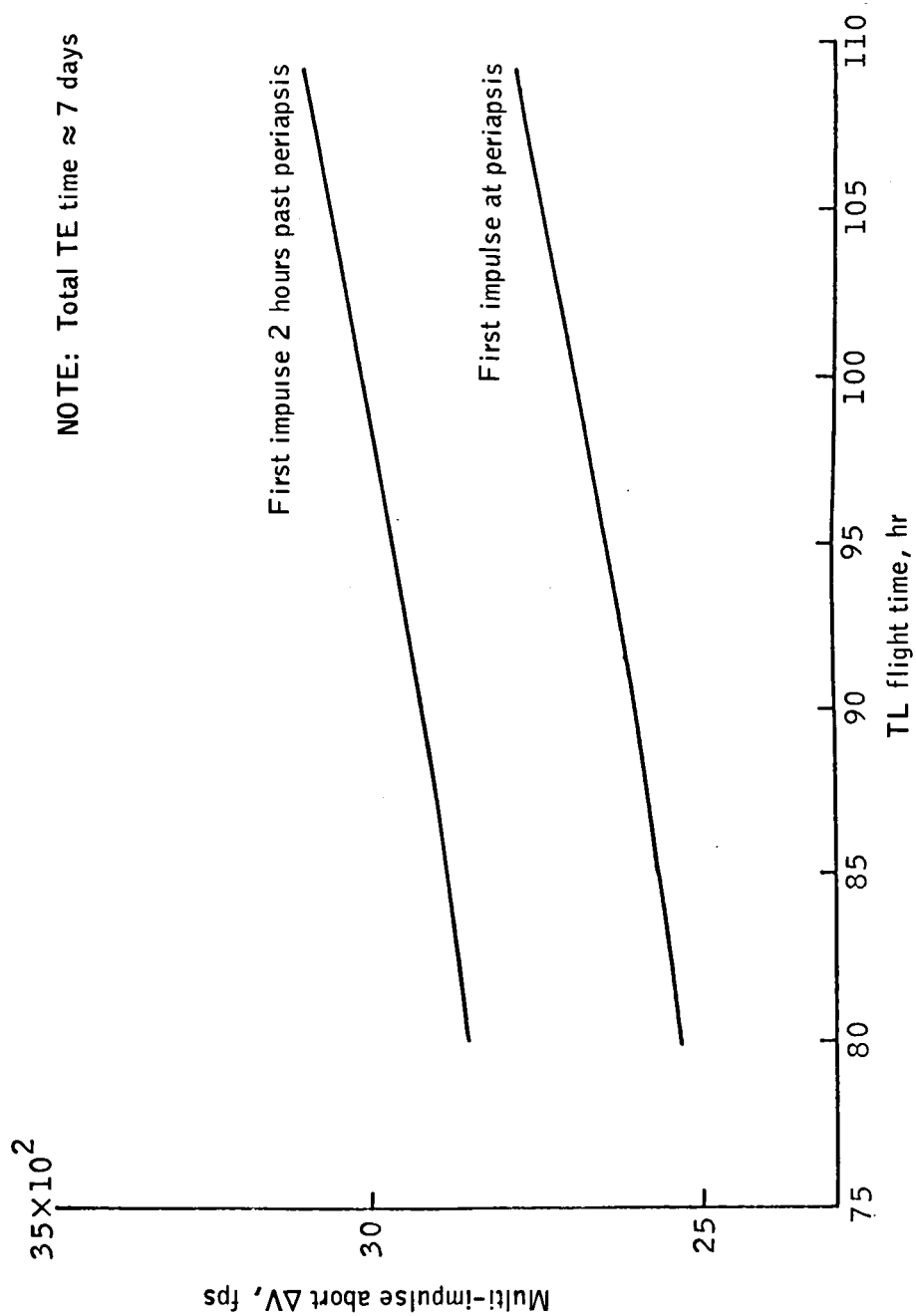


Figure 4. - Multi-impulse abort ΔV requirements of nonfree-return TL trajectories to Copernicus.

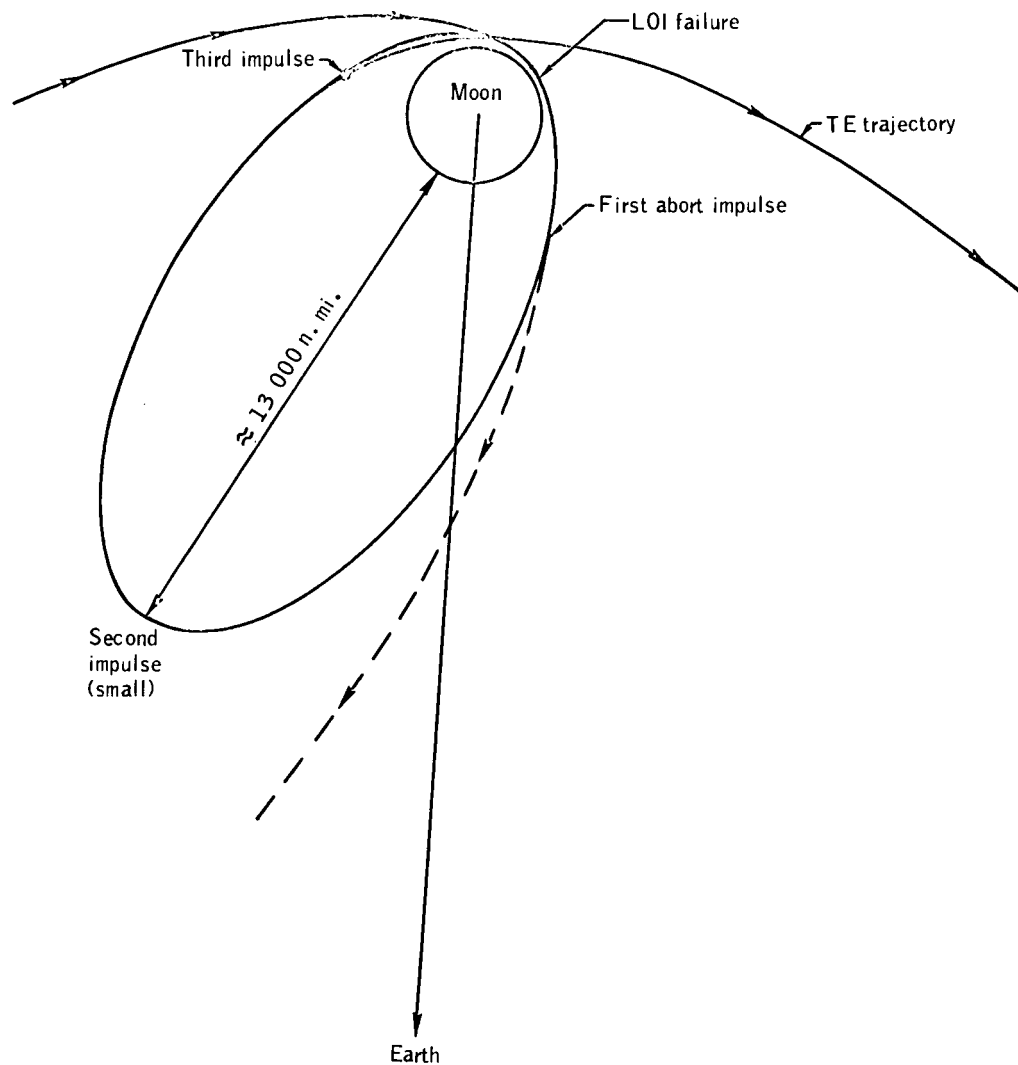


Figure 5. - Profile of multi-impulse abort after engine failure at LOI.

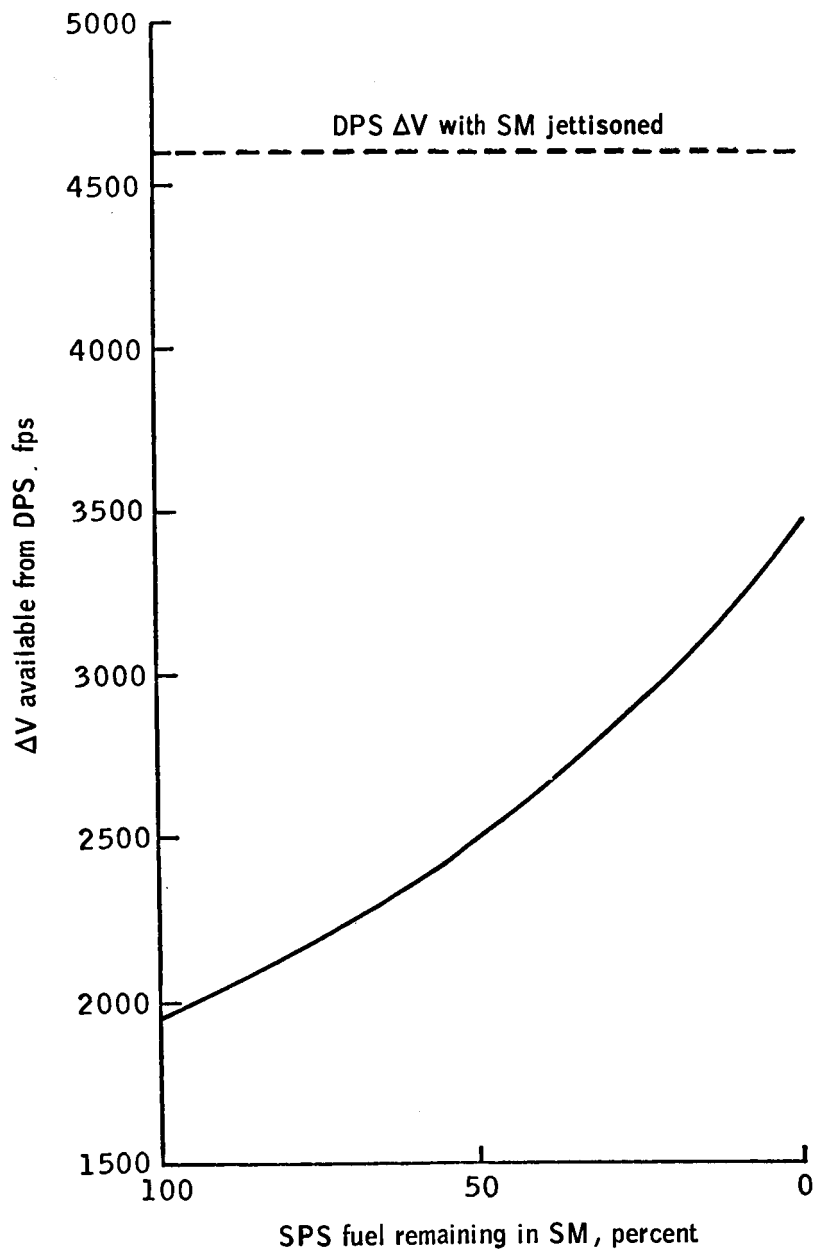


Figure 6. - Abort ΔV available from LM DPS.

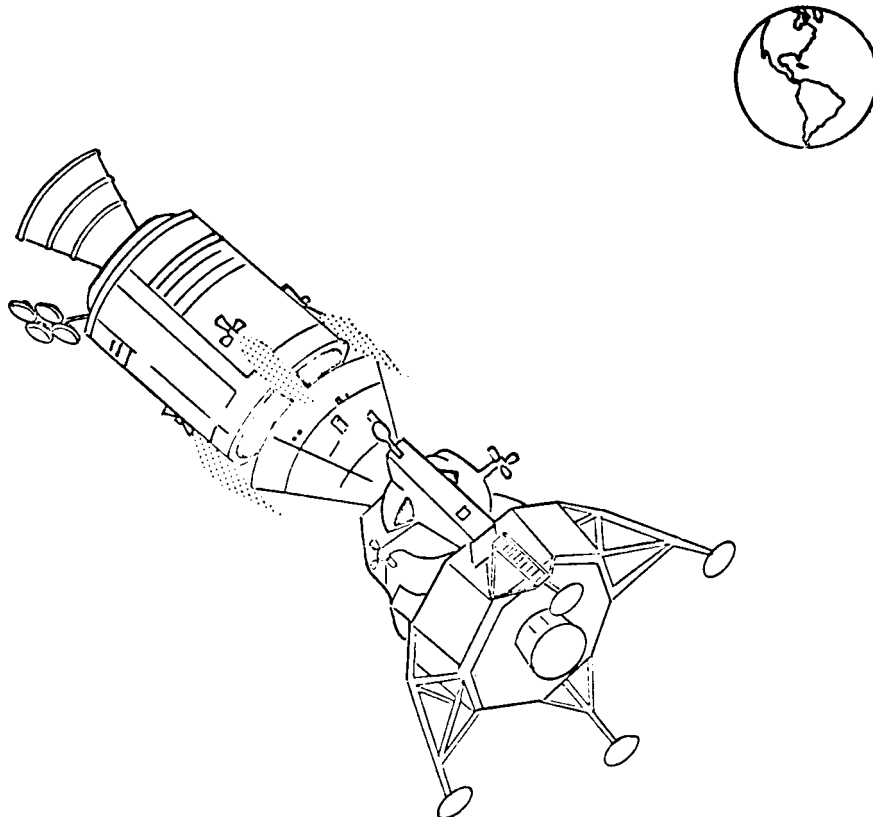


Figure 7. - Service-module separation prior to LM DPS abort propulsion maneuver.

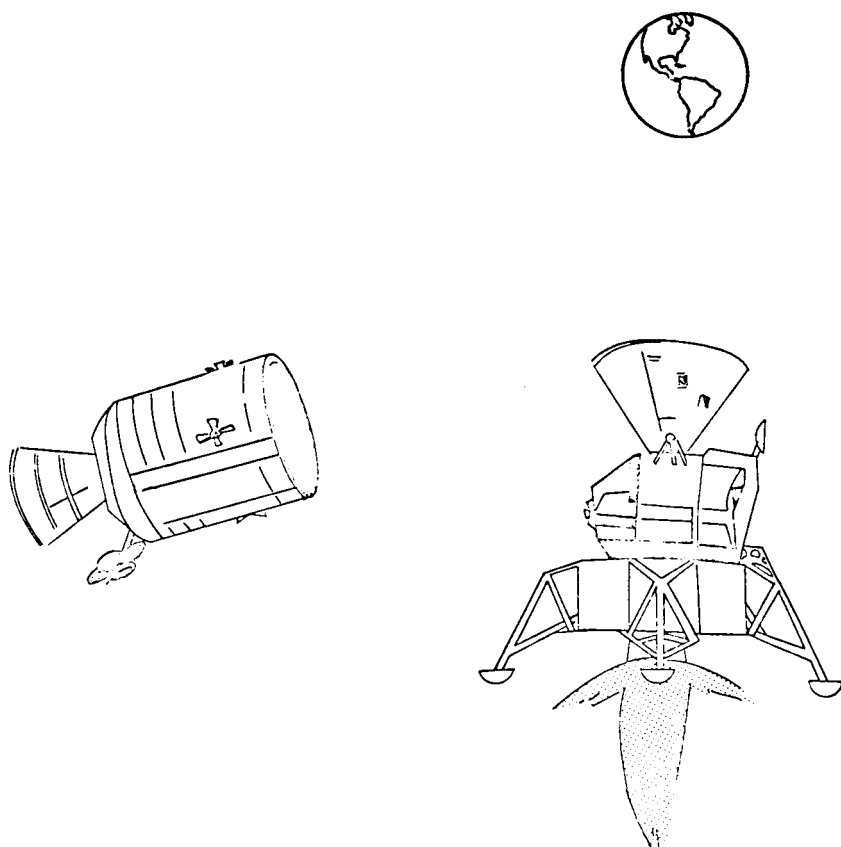


Figure 8. - Command-module DPS abort maneuver onto a high-speed TE trajectory following SM separation.

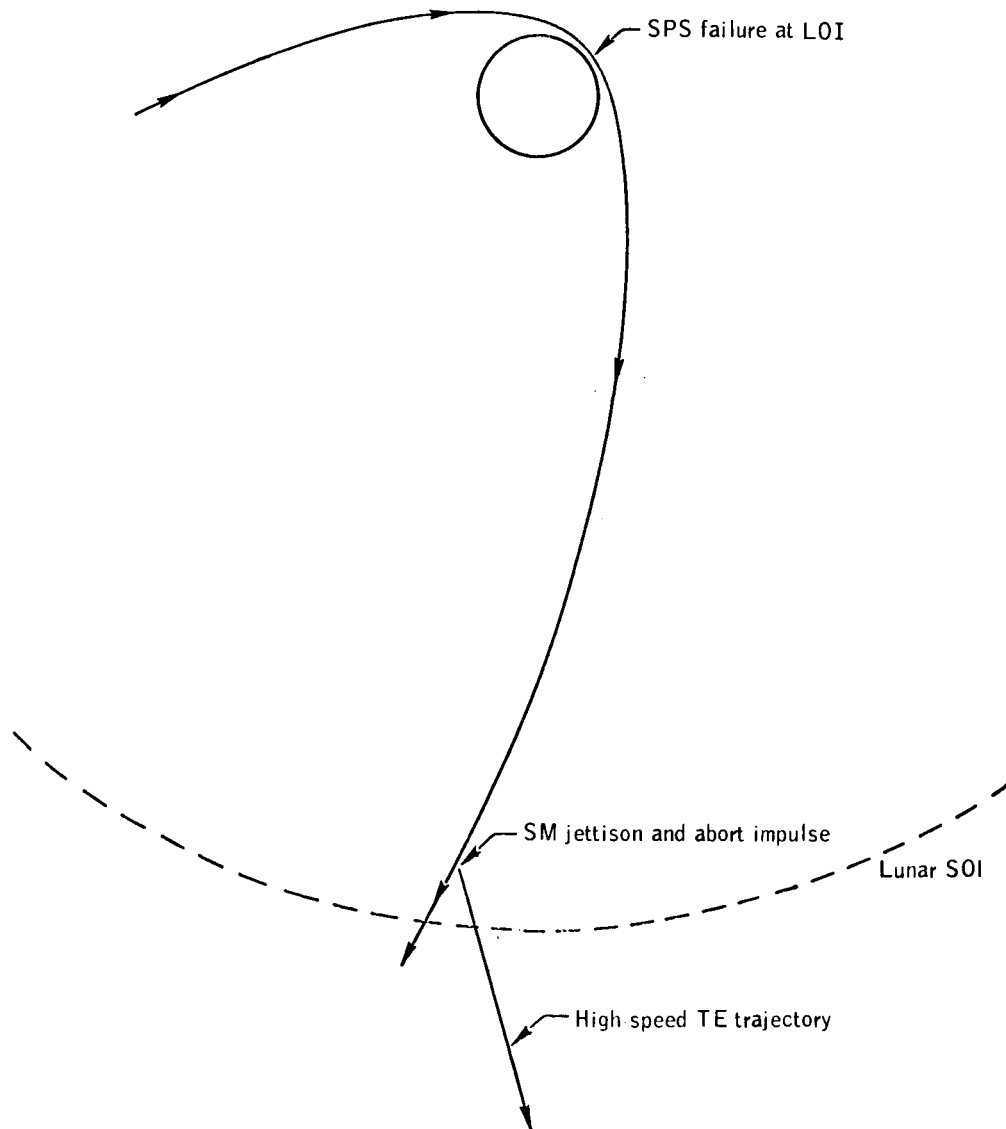


Figure 9. - Flight profile for abort with SM jettisoned to increase ΔV capabilities.

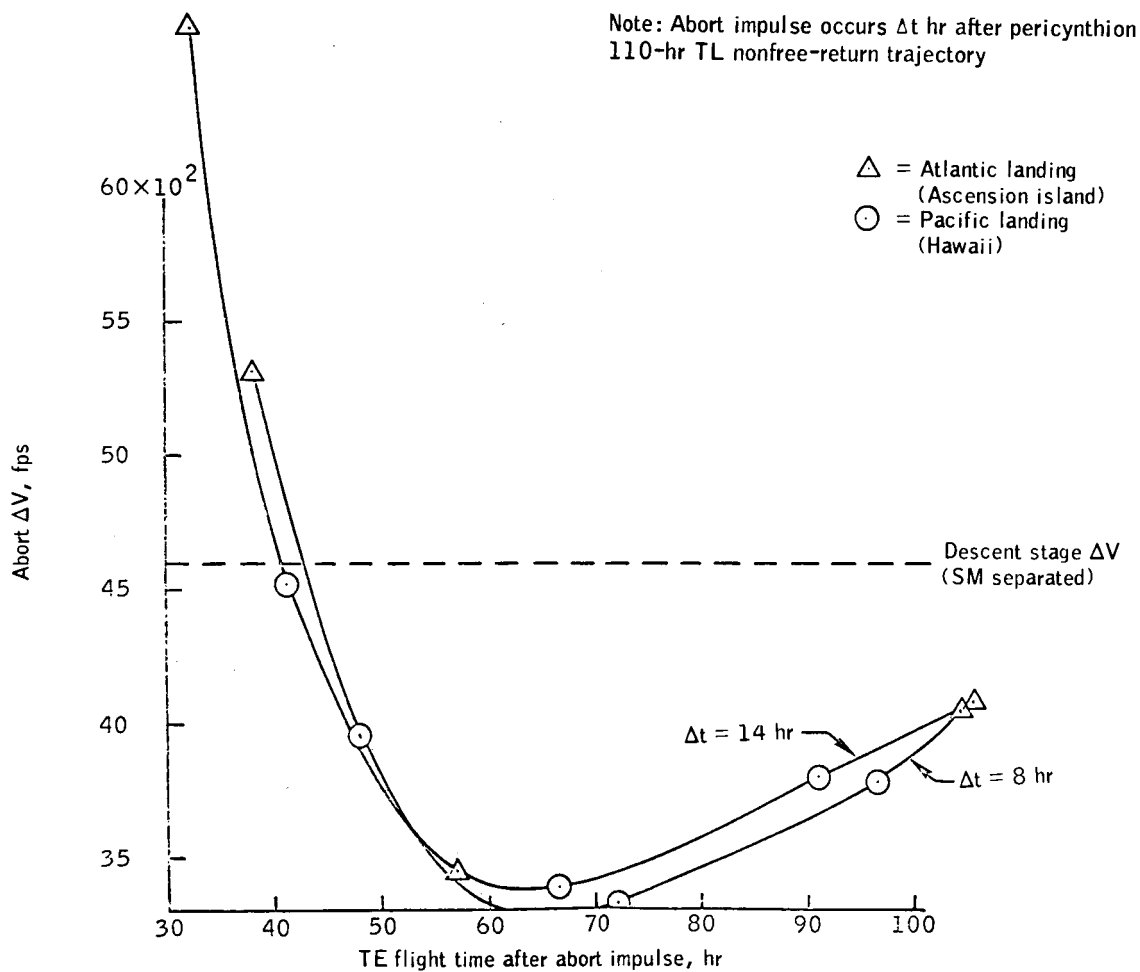


Figure 10. - Abort ΔV requirements for high-speed TE trajectories, Copernicus mission.

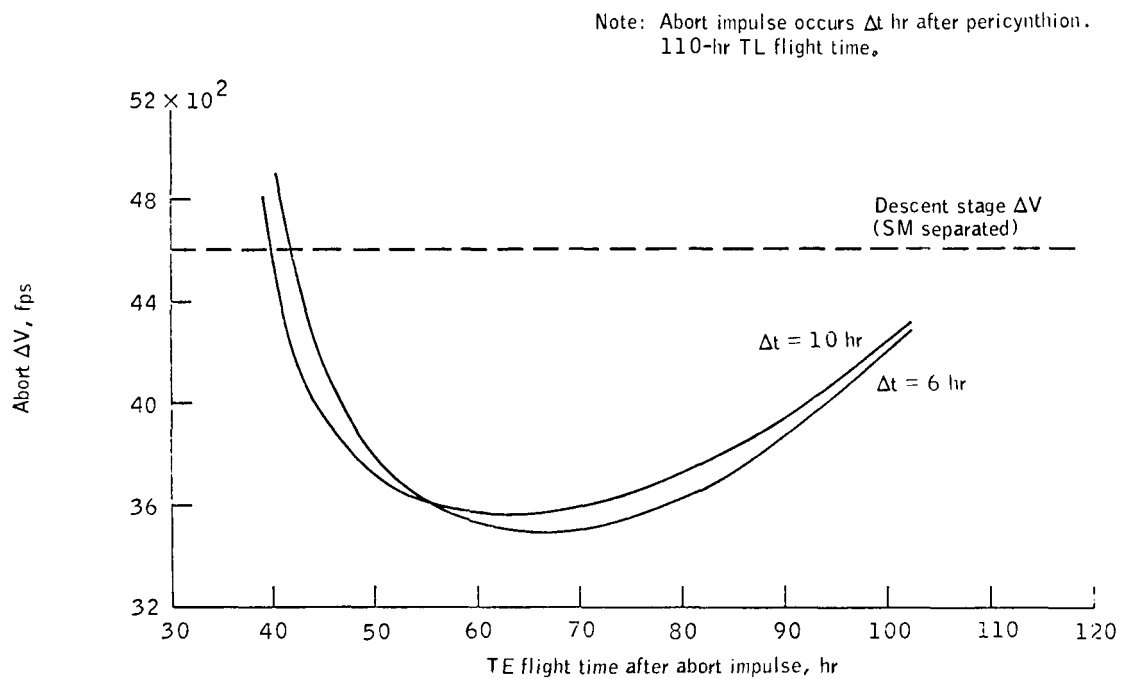


Figure 11. - Abort ΔV requirements for high-speed TE trajectories, Aristarchus mission.

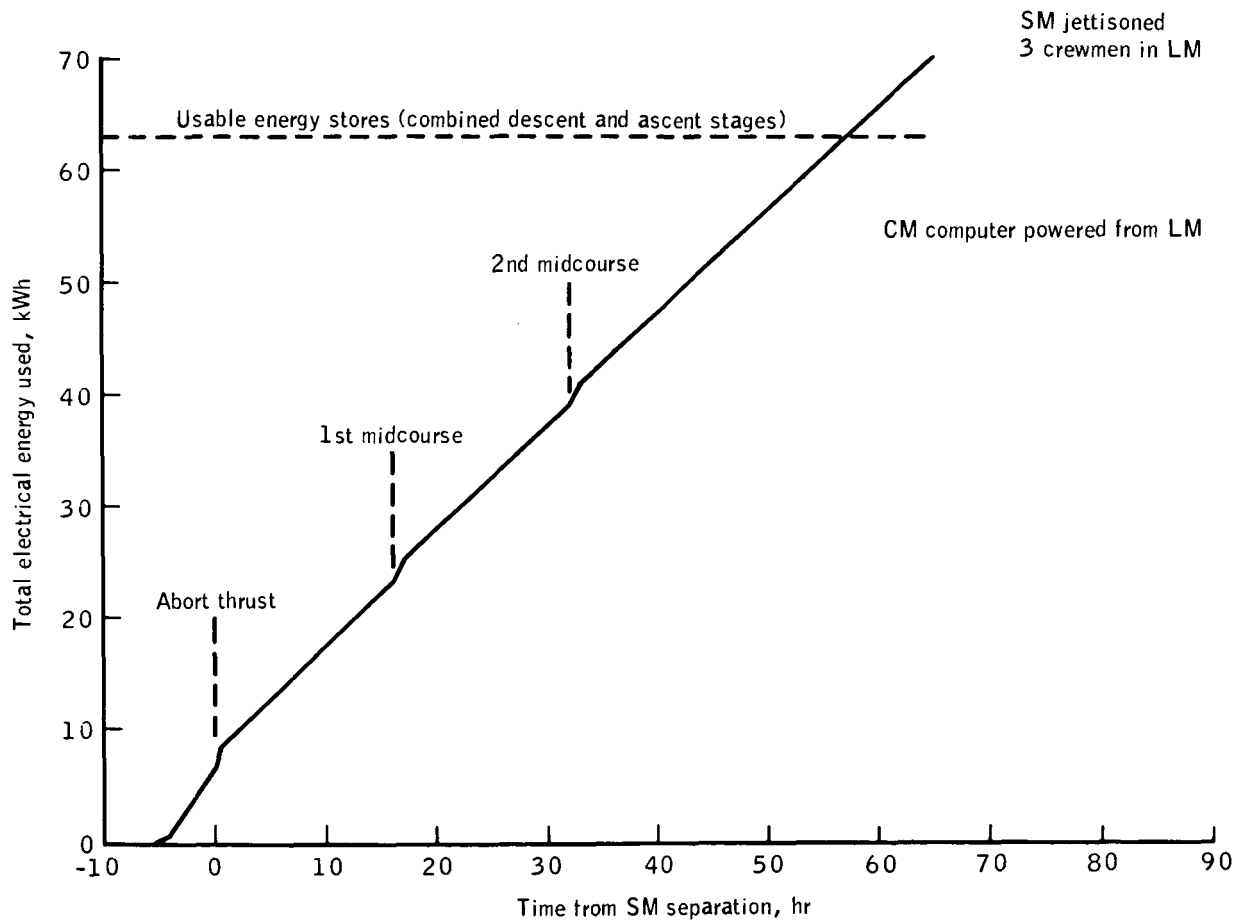


Figure 12. - Lunar-module electrical energy consumed to provide crew emergency life support after SM separation.

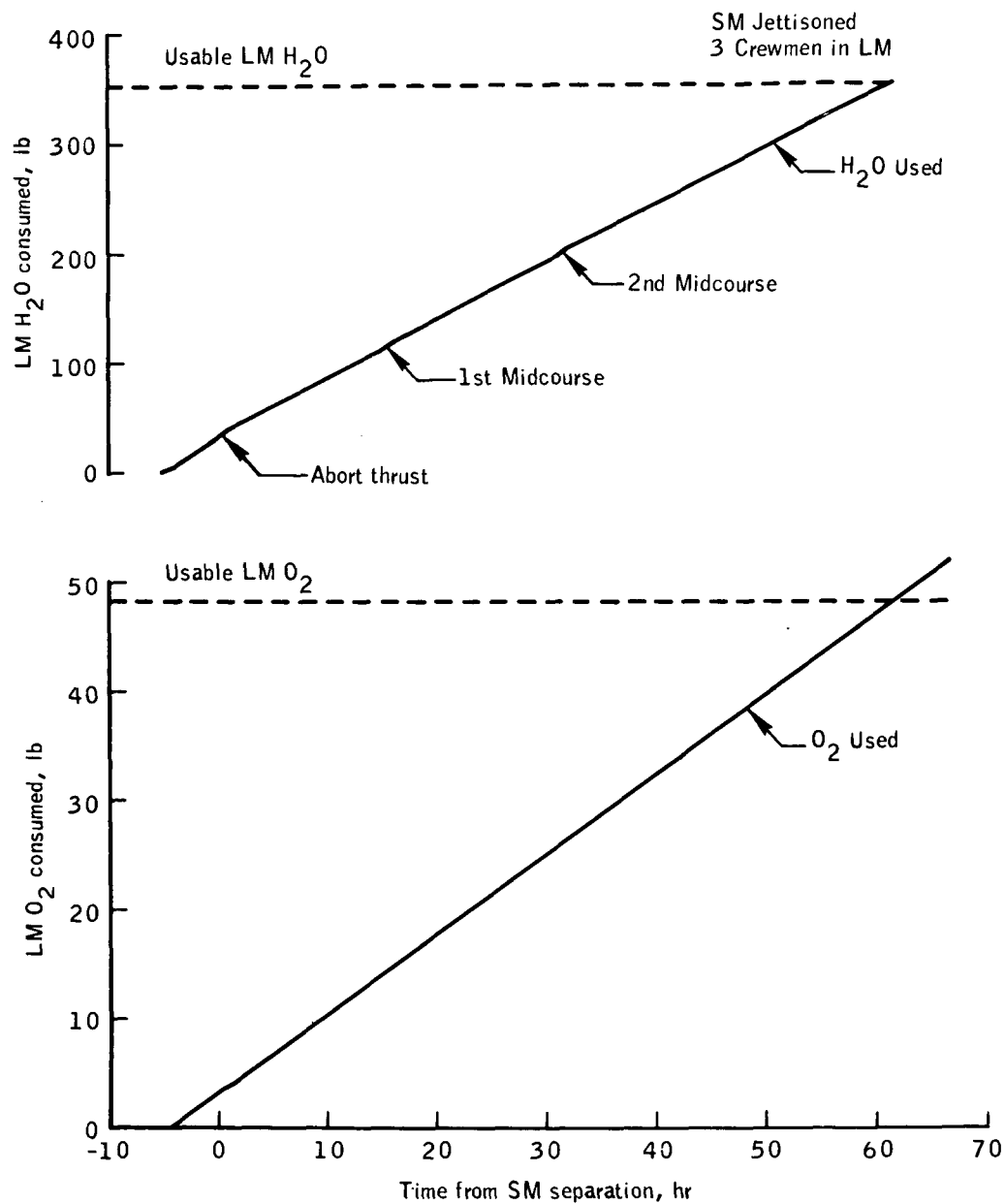


Figure 13. - Lunar-module O₂ and H₂O consumed to provide crew emergency life support after SM separation.